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SPACE SHUTTLE ENGINEERING AND OPERATIONS SUPPORT DESIGN NOTE NO. 1.2-DN-B0104-1 ADVANCED COMPOSITES - MECHANICAL PROPERTIES AND HARDWARE PROGRAMS FOR SELECTED RESIN MATRIX MATERIALS ENGINEERING SYSTEM ANALYSIS

ERWIN K. WELHART, ET AL

31 MARCH 1976

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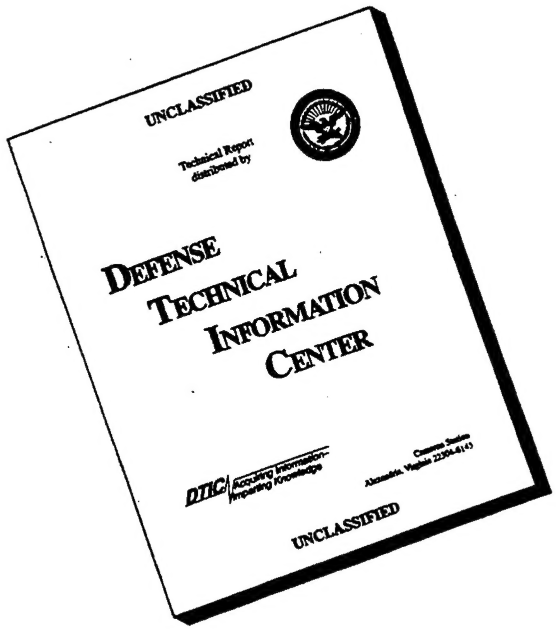

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SPACE SHUTTLE ENGINEERING AND OPERATIONS SUPPORT

DESIGN NOTE NO. 1.2-DN-B0104-1

ADVANCED COMPOSITES - MECHANICAL PROPERTIES AND HARDWARE PROGRAMS FOR SELECTED RESIN MATRIX MATERIALS

ENGINEERING SYSTEMS ANALYSIS

31 March 1976

This Design Note is Submitted to NASA Under Task Order No. B0404, In Partial Fulfillment of Contract NAS 9-13970.

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PREFACE

This report is the first of three reports planned to summarize the technology state-of-the art for graphite and boron reinforced epoxy and polyimide matrix materials. The reports are as follows:

1.2-DN-B0104-1
"Advanced Composites - Mechnaical Properties, and Hardware Programs for Selected Resin Matrix Materials"

1.2-DN-B0104-2
"Advanced Composites - Environmental Effects on Selected Resin Matrix Materials"

1.2-DN-B0104-3
"Advanced Composites - Fabrication Processes for Selected Resin Matrix Materials"

The data and information presented is intended as an adjunct to on-going NASA studies to determine the relative merits of using composites in the Space Shuttle Program and identify those design applications which are high value candidates for utilizing composite materials.

SUMMARY

This Design Note presents typical mechanical properties tabulated from Industrial and Governmental Agencies' test programs. All data are correlated to specific products and all of best known products are presented. The data includes six epoxies, eight polyimides and one polyquinoxaline matrix material. Boron and graphite are the fiber reinforcements.

Included in this Design Note are forty-two summaries of advanced (resin matrix) composite programs in existence in the United States. The extent of composite use, both experimental and production, indicates that both Government and commercial aerospace programs are considering various major parts (primary and secondary structure) of aircraft, missiles and spacecraft as candidates to be fabricated from advanced composites.

After studying the available data from selected reports (References 1 thru 23), it is concluded that the selection of appropriate matrices, the geometric manner in which the fibers are incorporated in the matrix and the durability of the bond between the fiber and the matrix establish the end properties of the composite material and the performance of the fabricated structure.

INTRODUCTION

This Design Note is intended to furnish designers, structures, and materials and processes engineers with advanced composites property data and experience descriptions to permit them to evaluate the appropriateness of these materials for Shuttle applications. Because of rapidly changing technology, data contained herein must be considered as only representative of composite materials. Use of the data for a specific design is cautioned without verification of appropriateness. This report documents typical mechanical properties of the most widely used resin matrix composites; a discussion of joints; and lists and describes briefly the major advanced composite programs currently in existence in the United States.

Graphite and boron reinforcement composites are considered direct competitors in many respects. Depending on the specific application, each material has its place. Objective consideration of the properties of alternative reinforcing materials can make the difference between a successful application and a costly flop. Further, having the right fiber for the job is not the whole story. The matrix causes the reinforcing fibers to work together and to sum their individual strengths and stiffnesses. The matrix material is generally the "weak link" in a composite application.

DISCUSSION - General

Incorporation of high-strength, high-modulus, and low density filaments into a compatible matrix generates a material which offers potential for major breakthoughs in aerospace vehicle design. These materials are classified as "advanced composites." The term "advanced composite" is specifically defined as, and limited to, composites characterized by high-strength, high-modulus fibers. Within the confines of this limitation, however there are no restrictions on either filament or matrix material, volume fraction, or filament orientation. Data presented in the Properties Section of this design note reflect the current status of material characterization for advanced composites. Other high strength, high modulus filament materials are under development, as well as more advanced matrix materials. As data on additional material systems become available, consideration will be given to having it incorporated into subsequent editions of this design note.

Information on the environmental effects on material properties and producibility factors will be presented in subsequent design notes (1.2-DN-B0104-2 and 1.2-DN-B0104-3). Including these effects will determine whether or not a realistic design is obtained.

The weight potential obtainable from advanced composite materials, such as boron and graphite reinforced epoxies and polyimides, cannot be overlooked for possible use in several Shuttle applications. The vast potential of advanced composites results from their four fold

increase in both stiffness and strength efficiencies. This increase allows design applications where advanced composite materials can be bonded directly to conventional structural metals such as aluminum and titanium with the assurance that, at ultimate load, each material is at or near its own ultimate stress level. Thus, the potential for high volume usage of advanced composites surpasses that of fiberglass laminates, which are very strong but not stiff, and beryllium, which is very stiff but not strong.

DISCUSSION - Joints

Advanced composites have the inherent potential to produce a weight saving of approximately 50 percent of the structural weight of a vehicle. However, not all this is generally realized. One of the reasons is the required design in the area of joints. Because of the nonhomogeneity of the composite material structure, it is desirable to avoid or at least mollify areas of stress concentration. Joints, which are present when any two components are assembled, are a source of stress concentrations. In the case of bonded joints, stress concentrations occur but the bond helps maintain strain compatibility between components. In the case of mechanical joints, stress concentrations are a result of the decreased area at the hole and the loaded hole itself. For composite materials the most efficient joints are scarf or stepped lap joints, in which there is relatively little change in the load path. Adhesive joints are more efficient for lightly loaded joints, while mechanically fastened joints are more efficient for highly loaded joints.

For maximum effectiveness and confidence, adhesive bonded joints should be designed in accordance with the following general principles:

(a) The bonded area should be as large as possible, within the allowable geometry and weight constraints.

(b) A maximum percentage of the bonded area should contribute to the strength of the joint.

(c) The adhesive should be stressed in the direction of its maximum strength (shear)

(d) Stress should be minimized in the direction in which the adhesive is weakest (tension or peel).

Mechanical joints require a mechanical fastening agent and are characterized by the cutout (hole) required for the fastener. Examples are riveted, pinned, and bolted joints. Because of the cutout, only a certain percentage of the ultimate material strength is generally available for design purposes. Since this is generally intolerable, various local reinforcing methods, such as metallic reinforcements, doublers, or local ply buildups, are used to develop acceptable joint strengths. For more information see Rockwell International Corp. report RI-73A01-Vol. 1, Jan. 1973. (Reference No. 1).

DISCUSSION - Current Advanced Composite Hardware Programs

This Design Note describes forty-two advanced composite programs in existence in the United States. Applications of advanced composite materials to aircraft systems are currently under evaluation with

regard to three distinct areas of utilization. Each one promises somewhat different payoffs, but all have definite advantages over current all-metal aircraft designs. The three areas of advanced composite utilizations are:

(a) Total composite designs.

(b) Element-by-element substitution of composites for metallic designs.

(c) Selected reinforcement of metal elements with composites.

Each summary attempts to outline the program objectives; system, component, or part investigated; pertinent test results; and conclusions. For more information on composite hardware programs see References 3 and 18.

MECHANICAL PROPERTIES - Introduction

Typical mechanical properties for graphite and boron fiber composites using epoxy and polyimide resin matrices are documented in this section. A graphite polyquinoxaline is also included. There are other high temperature resin matrix systems available and being developed for use with graphite and boron fibers such as polybenzimidazoles, polyarylsulfones and pyrrones but they are not discussed herein because of their advanced state-of-the art status.

Data were tabulated from 23 Government and Industry reports and covers the time period from 1969 thru 1974. The data tabulated were, in practically all cases, the averages of the data points recorded from the test programs. In some cases the reports stated that the strengths or moduli shown were calculated averages from "X" numbers of individual

data points. Rarely did a report show "A" or "B" design allowables and when they did, the raw data utilized was not included as part of the report.

MIL-HDBK-5B, in the section on procedure for calculating design allowables, states that 100 data points may be adequate to allow the determination of A and B values, provided the data are near normal distribution. If the distribution is not normal, at least 300 data points are required so that computation can proceed without knowledge of the distribution form. There were not enough data points collected for any one property to allow calculations of MIL-HDBK-5 method design allowables. Therefore, the data shown in the Properties Section are typical values and their use for specific designs would require appropriate reduction factors.

It is possible to fabricate a composite laminate with many different fiber volume fractions. Fiber volume is the amount of fiber present in a composite and is expressed as a percentage volume fraction or weight fraction of the composite. However, a great deal of the existent test data have been generated for one specific fiber volume fraction for each generic system. These volume fractions were selected based on theoretical micromechanical analysis which determined the most efficient configurations. All of the boron/epoxy data shown in the tables are for a 50-percent volume fraction (fiber reinforcement), while the data on graphite/epoxy reflects a 60-percent volume fraction. These fiber volume fractions are the ones most generally available in the off-the-shelf prepreg materials, although the graphite/epoxy volume

fraction may vary between 55 and 60-percent, depending on the supplier. Virtually all the actual hardware items built to date have been fabricated using these standard volume fractions.

Design requirements that can be met by boron or graphite fiber reinforced resin matrix composites are shown in Table I. Thermosetting resins such as epoxies and polyimides are used as matrix materials in advanced composites. Table II shows the general characteristics of the two most used thermosetting resin materials.

TABLE I DESIGN CONSIDERATIONS FOR ADVANCED COMPOSITES

BORON COMPOSITES

DESIGN REQUIREMENT	PROPERTY
High compression loading	High compressive strength per unit density (high specific compressive strength)
Large sheet goods; little or no machining	Large-diameter fiber; collimates well; forms straight, high-equality sheet
Combination of high stiffness and high strength	High specific (per unit) density, strength and modulus in same fiber
Minimum thermal distortion when combined with metals	Thermal coefficient of expansion for boron closer to metals (such as titanium) than graphite

GRAPHITE COMPOSITES *

DESIGN REQUIREMENT	PROPERTY
Thin-wall construction under low compressive loading	High specific (per unit density) modulus provides high buckling stability
Structures with cutouts, holes, or tapers	Easily machined by conventional metal-removal techniques
Sandwich structures; thin skins	Individual ply thickness can be varied from 0.001 to 0.010 in; ability to produce specific gages in laminate
Material must be formed around tight radius	Small fiber diameter permits small radius of curvature without incurring high stresses in the fiber
Exact modulus, or more than one modulus required in the same structure	Available in a range of moduli from 2 to 80 million psi*

*The strength of graphite fiber decreases as its modulus (stiffness) increases. On the other hand, modulus of graphite is controllable over a wide range, whereas that of boron is not. Graphite fiber can be produced in much higher moduli than boron if strength can be sacrificed.

TABLE II MATRIX SYSTEMS-GENERAL CHARACTERISTICS

Matrix Material	Maximum Service Temperature Range	General Characteristics
Modified epoxy	350°F continuous 420°F intermittent	Thermosetting resin utilized for low-pressure (approximately 100 psi) laminating requiring a minimum of a 350°F cure.
Polyimide	550F continuous 700°F intermittent	Thermosetting resin utilized for low-pressure (approximately 200 psi) laminating requiring 350° to 600°F cure plus an extended post-cure.

MECHANICAL PROPERTIES - Epoxy Matrix

The epoxy-based matrix materials are used in applications up to 350^{o}F; are resistant to most chemicals and solvents; are practically 100% reactive, therefore, do not release volatiles causing voids in the matrix; there is no significant shrinkage or expansion, therefore, design dimensions are easily met; and the resin is stronger than any other polymer except Kevlar 49 fiber material. It is epoxy matrix materials for which the greatest amount of data are available. Tables III and IV show the mechanical properties for nine graphite/epoxy unidirectional composites and one boron/epoxy unidirectional composite. For more information see References 1, 2, 3, 6, 8, 11, 12, 14 and 22.

MECHANICAL PROPERTIES - High Temperature Matrix

For the temperature range of 350^{o}F to 600^{o}F, polyimide resins are being used. They present greater fabrication problems, compared to epoxies, mainly because of outgassing during cure. They are available and usable materials, even though considerable development and data generation remains to be completed. The polyimide systems represent the current upper temperature limit for organic materials.

Tables IV, V and VI show the mechanical properties for four boron/polyimide unidirectional composites, fourteen graphite/polyimide composites and one graphite/polyquinoxaline composite. For more information see References 4, 12, and 14. Systems other then epoxy and polyimide are under development for even higher temperature ranges,

but are still in the experimental stage for use with advanced fibers. These polymers are: polybenzimidazoles, pyrrones, polyarylsulfones, and polyphenylene sulfide.

TABLE III
MECHANICAL PROPERTIES OF GRAPHITE/EPOXY COMPOSITES
(TYPICAL VALUES)

MATERIAL / TRADE NAME / COMPANY — MECHANICAL PROPERTIES		GRAPHITE/EPOXY AS/SP286 3M CO.	GRAPHITE/EPOXY AS/3002 HERCULES/ FIBERITE	GRAPHITE/EPOXY T300/5208 WHITTAKER	GRAPHITE/EPOXY HTS/SP286 3M CO.	GRAPHITE/EPOXY AS/3501 HERCULES INC.	GRAPHITE/EPOXY TYPE II/5206 WHITTAKER	GRAPHITE/EPOXY TYPE III/5206 WHITTAKER	GRAPHITE/EPOXY HMS/X904 FIBERITE	GRAPHITE/EP(T300/934 FIBERITE *
FIBER ORIENTATION		0°	0°	0°	0°	0°	0°	0°	0°	.007 FABR
0° FLEX STRENGTH (Ksi)	77F	207.3	233.9	271.5	205	224	197	173	121.5	
	250F									
	350F	146.5	188.3	210.0	141	154	139	121	77.3	
0° FLEX MODULUS (Msi)	77F	16.4	15.9	22	17	20.1	20.	15.3	22.1	
	350F	15.9	16.0		16.4		19.2	14.4	19.9	
0° TENSILE STRENGTH (Ksi)	-67F	162.0								
	77F	194.4	239.8	247	199.3		137.3	157.7	92.5	58.5
	250F	202.0								
	350F	181.6	194.9	235	142.8		146.3	140.8	83.7	45.6
0° TENSILE MODULUS (Msi)	-67	16.4								
	77F	17.9	17.25	23.3	19.62		21.59	17.32	29.1	9.78
	250F	18.2								
	350F	17.8	16.4	22.9	18.83		20.92	16.16	32.0	9.4
0° TENSILE STRAIN (μ in / in)	77F	9323	13477		9767		6195	8989		
	350F	11,361			7461		6773	8674		
0° COMP. STRENGTH (Ksi)	77F	171	256.4	275.5	226.6		205	150.3	83.6	43.9
	250F	163								
	350F	123	75.7	197.6	119.7		84.3	86.0	57.3	36.0
0° COMP. MODULUS (Msi)	77F	16.5								
	250F	15.3								
	350F	15.9								
0° COMP. STRAIN (μ in/in)	77F	15256	19661	17645	15232		11420	10003		
	350F	5787	8348	11360	7497		4501	5897		
90° TENSILE STRENGTH (Psi)	-67F	11600								
	77F	9880	8637	10190	8516		8098	7740	3696	58.5
	250F	7800								
	350F	6329	4043	7050					2500	45.6
90° TENSILE MODULUS (Msi)	-67F	2.17								
	77F	2.1	1.97		1.96		1.72	1.92	0.87	9.58
	250F	1.51								
	350F	1.13								10.1
90° TENSILE STRAIN	77F	5792	5383	5120	4535		4921	4268		
	350F			5387						
HORIZONTAL SHEAR STRENGTH (Ksi)	77F	14.36	15.0			16.7			10.5	8.1
	350F	6.84	8.25							2.6
INTERLAMINAR SHEAR STRENGTH (Psi)	77F	14,500	13725	17800	15100		15500	14600	9800	
	350F	6,900	9206		6900		7100	6200	5130	
90° COMP. STRENGTH (Ksi)	77F	45.6								43.9
	250F	29.1								
	350F	19.9								36.0
90° COMP. MODULUS (Msi)	77F	2.9								
	250F	2.1								
	350F	1.8								
SHORT BEAM SHEAR (Ksi)	77F			12.7		17.4				
	350F			7.2		8.6				
90° FLEX STRENGTH (Ksi)	77F		11.6							

* FIBERITE HMF 3308 FABRIC

TABLE IV
MECHANICAL PROPERTIES OF BORON/POLYIMIDE AND EPOXY COMPOSITES (TYPICAL VALUES)

MECHANICAL PROPERTIES \ MATERIAL TRADE NAME CO.		BORON/POLYIMIDE BORON/SB703 MONSANTO	BORON/POLYIMIDE BORON/SB709 MONSANTO	BORON/POLYIMIDE BORON/P13N TRW	BORON/POLYIMIDE BORON/4707 DUPONT	BORON/EPOXY BORON/SP296 3M Co.
FIBER DIRECTION		0°	0°	0°	0°	0°
0° FLEX STRENGTH (Ksi)	77°F 550°F	210.8 134.1	192.4 148.0	159.1 135.8	193.4 196.8	
0° COMP. STRENGTH (Ksi)	77°F	365.7				140 (350°F)
0° COMP. MODULUS (Msi)	77°F	32.7				
90° FLEX. STRENGTH (Ksi)	77°F 550°F	8.4 5.1	6.1 5.8	8.3 5.1	6.3 4.2	
90° COMP. STRENGTH (Ksi)	77°F	20.8				
90° COMP. MODULUS (Msi)	77°F	2.44				
0° COMP. STRAIN FAIL (μ in/in)	77°F	11811				
90° COMP. STRAIN FAIL (μ in/in)	77°F	11538				
SHORT BEAM SHEAR (Ksi)	77°F 550°F	9.75 10.6	3.8 4.2	5.6 5.3	7.9 7.8	
0° TENSILE STRENGTH (Ksi)	77°F 350°F					210 180
0° TENSILE MODULUS (Msi)	77°F					30
90° TENSILE STRENGTH (Ksi)	77°F 350°F					11 6.25
HORIZONTAL SHEAR (Ksi)	77°F 350°F 420°F					13 5 4
90° TENSILE STRAIN (μ in/in)	77°F					4100

TABLE V
MECHANICAL PROPERTIES OF GRAPHITE/POLYIMIDE COMPOSITES (TYPICAL VALUES)

MATERIAL TRADE NAME CO. / MECHANICAL PROPERTIES	GRAPHITE/ POLYIMIDE MODMOR II/SB703 MONSANTO	GRAPHITE/ POLYIMIDE MODMOR II/SB703 MONSANTO	GRAPHITE/ POLYIMIDE MODMOR II/SB703 MONSANTO	GRAPHITE/ POLYIMIDE MODMOR II/SBRS-6234 MONSANTO	GRAPHITE/ POLYIMIDE HM/SB703 MONSANTO	GRAPHITE/ POLYIMIDE MORGANITE II/SB 703 MONSANTO	GRAPHITE/ POLYIMIDE COURTAULDS HTS/ SB RS-6234	GRAPHITE/ POLYIMIDE HMS/ SB 710 MONSANTO	GRAPHITE/ POLYIMIDE THORNEL 50/SB 7 MONSANT
LAY-UP DIRECTION	0°	0° - 90°	±45°	0°	0°	0°	0°		
0° FLEX STRENGTH 77°F	165.45				81.4	147	206	110	113.8
600°F					33.7	62.5	146	85.6	113.8
0° FLEX. MODULUS (Msi) 77°F	17				21.7				
600°F					12.3				14.6
0° TENSILE STRENGTH (Ksi) 77°F	187			174.4	190		173.2		
550°F	104			109.3			161		
0° TENSILE MODULUS 77°F (Msi)	20.1			20.5					
0° COMP. STRENGTH 77°F (Ksi)	101.9								
0° COMP. MODULUS 77°F (Msi)	16.9								
90° FLEX. STRENGTH 77°F (Ksi)	7.1								
90°F TENSILE STRENGTH (Ksi) 77°F	7.26								
90° TENSILE MODULUS (Msi) 77°F	1.38								
RAIL SHEAR MODULUS (Msi) 77°F		10.24	38.5						
550°F		6.27	21.5						
RAIL SHEAR MODULUS (Ksi) 77°F		.721	6.4						
550°F		.436							
SHORT BEAM SHEAR 77 F								4.59	
600 F								3.58	

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TABLE VI
MECHANICAL PROPERTIES OF GRAPHITE/POLYIMIDE AND POLYQUINOXALINE COMPOSITES (TYPICAL VALUES)

MATERIAL TRADE NAME / MECHANICAL PROPERTIES	GRAPHITE/ POLYIMIDE HM/P13N	GRAPHITE POLYIMIDE HT/P13N	GRAPHITE/ POLYIMIDE MORGANITE II/ P13N	GRAPHITE/ POLYIMIDE THORNEL 50/ P13N	GRAPHITE/ POLYIMIDE HM/PYRALIN 4707	GRAPHITE/ POLYIMIDE MORGANITE II/ GEMON	MORGANITE II/ POLYQUINOXALINE
LAYUP DIRECTION	0°	0°	0°	0°	0°	0°	0°
0° FLEX. STRENGTH (Ksi) 77°F	147	225	242	107	40.5	170	106.6
600°F	82.4	110	101	85	32.7		99.7
0° FLEX. MODULUS (Msi) 77°F	25.5	23	15.3	14.9	23.9	15.3	15.5
600°F	25.5	22			21.2		15.1
90° TENSILE STRENGTH (Ksi) 77°F	9.9						

MECHANICAL PROPERTIES -

Typical Fiber Properties and Trade Names

Graphite and boron reinforcements are considered direct competitors in many respects. Depending on the specific application and fabrication problems each material has its place. The only variation obtainable in boron fibers is the fiber diameter. Boron fiber tape and broadgoods prepregs can be purchased with boron fibers having diameters of 0.004 or 0.0056 inches. There is a weight saving advantage when using the 0.0056 inch diameter fibers. The tungsten center is the same diameter in both fibers, therefore the overall density is less for the larger fibers. Graphite materials are available in several types ranging from low modulus to ultra high modulus properties. As the modulus increases the tensile strength decreases. This is due to the fact that the modulus increase is a function of the percentage of fiber graphitization. Crystalline graphite has a higher modulus than amophorous carbon. By the same token, as the percentage of graphitization increases, the fiber tensile strength is more dependent upon the bond strength between crystals, which is less than the tensile strength of amophorous carbon. Tables VII and VIII show the typical properties of all the types and the vendor trade names for all types respectively.

TABLE VII TYPICAL GRAPHITE FIBER PROPERTIES

	Type A-S	Type HT-S	Type HM-S	Type GY-70
E msi	25 - 32	32 to 40	50 - 60	70 - 83
UTS, ksi	400 to 450	350 to 400	250 to 350	220 to 300
ρ. lb/in^3	0.0630	0.0650	0.0703	0.070

TABLE VIII GRAPHITE FIBERS
CLASSIFIED BY TRADE NAMES

25-32 msi LOW MODULUS, LOW TO ULTRA HIGH STRENGTH	32-40 msi MEDIUM MODULUS, HIGH STRENGTH	50-60 msi HIGH MODULUS, MEDIUM STRENGTH	>70 msi ULTRA HIGH MODULUS, MEDIUM STRENGTH
Hercules Type A-S Stackpole Panex 30/A Courtaulds Type A Morganite Type III Union Carbide Thornel 300 Great Lakes Fortafil 3-T Thornel 400 Polycarbon A	Hercules HT-S Courtaulds HT-S Morganite II Modmor Type II Fortafil 4-T Polycarbon T Thornel 300	Hercules HM-S Courtaulds HM-S Morganite I Modmor Type I Fortafil 5-T Fortafil 6-T Thornel 50 Hitco HMG-50 Polycarbon M	Celanese Celion GY-70 Thornel 75

TABLE IX

COMPOSITES COMPARISON

PROPERTY DATA

MATERIAL DESIGNATION	SCOTCHPLY 1009 - 26S	SCOTCHPLY 1009 - 26C	KEVLAR - 49	5208/T300	5505/4	BORON/ALUMINUM	ALUMINUM (2024 - T81)
VENDOR	3M	3M	DUPONT	NARMCO	AVCO	AMERCOM	——
FIBER ORIENTATION	UNIDIRECTIONAL	UNIDIRECTIONAL	UNIDIRECTIONAL	UNIDIRECTIONAL	UNIDIRECTIONAL	UNIDIRECTIONAL	——
FIBER/RESIN	S-GLASS/EPOXY	E-GLASS/EPOXY	ORGANIC/EPOXY	GRAPHITE/EPOXY	BORON/EPOXY	——	(ALUMINUM SHEET)
TENSILE STRENGTH (PSI)	220,000	185,000	175,000	247,000	192,000	161,000	64,000
TENSILE MODULUS (10^6)	8.7	5.7	13	23.5	30	30	10
FLEXURAL STRENGTH (PSI)	230,000	200,000	95,000	295,000	——	——	——
FLEXURAL MODULUS (10^6)	7.9	7.0	11	22	——	——	——
COMPRESSIVE STRENGTH (PSI)	120,000	90,000	40,000	264,000	353,000	343,000	64,000
INTERLAMINAR SHEAR STRENGTH (PSI)	9,000	8,000	12,000 (1)	17,800 (1)	15,000	10,000	38,000 (SHEAR)
COEFFICIENT OF THERMAL EXPANSION (IN/IN/°F)	3.5×10^{-6}	1.1×10^{-6}	1.1×10^{-6}	$-.25 \times 10^{-6}$ (L) 17.4×10^{-6} (T)	2.3×10^{-6} (L) 10.6×10^{-6} (T)	4.0×10^{-6} (L) 12.7×10^{-6} (T)	13.3×10^{-6}
DENSITY (LB/IN^3)	0.072	0.072	0.050	0.058	0.0725	0.095	0.100
PLY/SHEET THICKNESS(IN)	0.0075	0.0075	0.0045 - 0.013	0.0055	0.0052	0.0072	.010-.012-.016
STOCK WIDTH (IN)	$\leq$ 12	$\leq$ 12	$\leq$ 12	$\leq$ 12	$\leq$ 12	$\leq$ 20	36 - 48
UPPER TEMPERATURE RANGE (°F)	300	300	350	400	420	600	350
MATERIAL COST ($/YD^2)	7.50 (2)	2.80 (2)	9.00 (2)	35.00 (2)	54.00	135.00	1.30/LB.

(1) SHORT BEAM SHEAR

(2) BASED ON 100 SQ.YDS.

CURRENT ADVANCED COMPOSITE HARDWARE PROGRAMS

A major reason for the United States' leadership in aerospace technology is the enormous research sponsored by numerous government agencies to develop superior materials and fabrication processes. The following is a summary of current research efforts sponsored by five government agencies: Air Force Materials Laboratory, Air Force Aero Propulsion Laboratory, Air Force Flight Dynamics Laboratory, National Aeronautics and Space Administration and Naval Air Systems Command. The extent of composite use, both experimental and production, indicates that both Government and commercial aerospace programs are considering various major parts (primary and secondary structure) of aircraft, missiles and spacecraft as candidates to be fabricated from advanced composites as evidenced by the following abstracts, the majority of which were summarized from Reference 18.

Aircraft Systems Applications

Boeing Aircraft Co.

737 Composite Spoilers - Phase I - Graphite/Epoxy
Phase II - Advanced Design Using Mixed Fibers

The flight spoilers of the 737 Airplane were selected as components to advance the use of Graphite/Epoxy materials for structural applications. Spoilers were developed with the aluminum skins replaced by Graphite/Epoxy and the aluminum end ribs replaced by fiberglass. The Graphite/Epoxy spoiler is 15 percent lighter than the all aluminum production spoiler. Plans are to fabricate and install 108 Graphite/Epoxy spoilers on 27 aircraft for flight service throughout the world. Phase II objective is to arrive at a more advanced version of the spoiler with maximum effective utilization of composite materials and a core such as Nomex or PRD-49 to replace the aluminum honeycomb core.

GENERAL DYNAMICS

F-111 Horizontal Stabilizer - Boron/Epoxy Skins on Al Honeycomb

A full-depth sandwich was chosen for design development. Major factors which led to the selection were weight, complexity comparisons and consideration of the previous development experience with the scaled version of the component. The component consisted of boron/epoxy skins; fiberglass spars and honeycomb core; and titanium root rib, pivot fitting and tip rib. Both static and fatigue test components were built. Static failure occurred in the attachment area in the region of the pivot fitting at a load of 136 percent of the limit design stresses. The primary cause for failure was found to be a design deficiency in the aft spar to hub fitting joint.
In the fatigue test, four lifetimes of the simulated F-111 horizontal tail loading spectrum were successfully applied.

Advanced Composites Technology - Fuselage Components — Boron/Epoxy. Graphite/Epoxy Boron/Aluminum Boron/high-temperature-plastic and molded plastic matrix composites

The component selected to be built first in this program was an F-111 aft fuselage section, 160 inches long, 48 inches deep and 36 inches wide. The fuselage component was primarily of graphite/epoxy construction, but also incorporated boron/epoxy and boron/aluminum in both tape and molded forms. A comparison of the composite test section with the existing metal section reveals that the application of composite material results in a weight saving of 19 percent.

For the second component of the fuselage program, the F-5 center section was chosen. A 26 percent weight reduction was achieved with this component.

GENERAL DYNAMICS/CONVAIR

Composite Configured Light Weight Fighter Aircraft

A 160-inch-long test component was fabricated and tested to destruction. Composite material systems of graphite/epoxy, boron/epoxy, boron/aluminum and glass/epoxy were employed. Maximum use of composites was used throughout the structure, with both skins and substructure being considered. A static test component was detail-designed and fabricated. Fourteen subassemblies of frames and panels were fabricated and assembled into a 920-pound component using 460 pounds of composite material. A weight saving of 18 percent of the counterpart aluminum structure was obtained. Static test of the part included loading in bending with simultaneous application of internal pressure. The component sustained 130 percent of design limit load.

Wing and Empennage-to-Fuselage Attachment Fittings - All advanced metal and resin matrix composites

The first phase is to establish the feasibility, extent and design concepts for using advanced composites for attachment fittings. The second phase will demonstrate this capability through full-scale static and fatigue tests of a typical wing root joint. The wing root joint used in the McDonnell Douglas F-15, a high performance jet-fighter, has been selected as typical of many similar aircraft attachments. Two subcontractors were chosen to support the development of two fitting types: Goodyear Aerospace, utilizing its experience with compression molded composites, to develop graphite/epoxy channel type bathtub fittings, and the Harvey Aluminum Co. to investigate composite reinforced forgings.

GRUMMAN AEROSPACE CORP.

F-14 Horizontal Stabilizer - Boron Epoxy

This program designed, fabricated, and static tested an all movable horizontal stabilizer utilizing boron/epoxy material for skins and portions of the substructure.

F-14 Airplanes with boron horizontal stabilizers are now in squadron service at Miramar Naval Air Station, Calif. The weight saving is 182 pounds per Aircraft.

Composite Wing Box Beam - Graphite/Epoxy, Boron/Epoxy and Boron/Aluminum

Program included the design, analysis, material characterization, process development, component fabrication and test. The component successfully completed both static and fatigue tests.

LOCKHEED AIRCRAFT CORP.

C-5A Right Hand Inboard Leading Edge Slats - 1 Static Test
1 Fatigue Test
1 Accelerated Service Test
10 Production Operational Aircraft

Boron/Epoxy

This program includes the design, fabrication and test of three inboard leading edge slats and incorporation of 10 production slats on operational aircraft. The three test articles were utilized in a static ultimate strength test, a fatigue test and an accelerated service test.

The slat has a platform area of 65 sq. feet and a total weight of 190 lbs., 21 percent less than that of the aluminum slat it replaces.

During static test ultimate loading was maintained for 80 seconds, after which a failure occurred on the outboard end of the slat. The second slat was installed on a C-5 aircraft and flight tested with no problems. The slat was then installed on the accelerated service test aircraft for an operational exposure period of 6 months.

The third slat is undergoing tests to demonstrate fatigue endurance under cyclic loading. These tests will extend to a span equivalent to four aircraft lifetimes or 120,000 flight hours.

C-130E Center Wing Box (3) 1 test, 2 flight - Boron/Epoxy

Three conventional center-wing boxes of the C-130E design will be fabricated and reinforced in the spanwise direction with unidirectional Boron/Epoxy composite. About 500 lbs. of Boron/Epoxy will be bonded to the skins and stringers of each wing box to reduce stress levels and increase fatigue life. The aluminum wing box weighs 4,800 lbs. and the Boron/Epoxy weighs only 4,200 lbs. Boron/Epoxy reinforced box must have a fatigue life of 40,000 hours. One panel was tested for six lifetimes and had a residual static strength of 109 percent limit load.

L-1011 Fairing Panels - PRD-49 Fiber/Epoxy Composites/Nomex H/C Core.

The PRD-49 fairing panels are to be service-tested on TWA, Eastern Air Lines and Air Canada Aircraft to represent a variety of route structures and service conditions. Test will be a 5 year service evaluation.
The component configurations for fabrication consist of seven wing-to-body honeycomb sandwich panels (one for static testing and six for installation on aircraft), six wing-to-body laminate filled panels, and six center engine honeycomb fairing panels, each approximately 60 X 80 inches in size.

The static test part for the wing-body fairing has been fabricated and successfully tested.

LTV-VOUGHT AERONAUTICS

A-7 Speed Brake - Graphite/Epoxy

The A-7 speed brake was chosen as a demonstration component for composite materials because it is large and highly loaded, but is not primary structure. The speed brake is about 9 ft. long and 6 ft. wide with the chines extended. The total predicted weight of the graphite/epoxy speed brake is 73.9 pounds compared to 123.4 pounds for the metal speed brake. This represents a 40 percent weight saving.

MCDONNELL DOUGLAS CORP.

F-4 Boron Rudder - Boron/Epoxy

Program was to demonstrate the feasibility of Boron/Epoxy composite usage in flight structures. Fifty rudders were built and have been in service for approximately 2 years. The Boron/Epoxy rudder is 35 percent lighter than the aluminum counterpart. The rudder has been successfully static tested to 310 percent design limit load before failure.

F-15 Composite Wing - 9 Ribs
1 Spar
All Hat and Tee Stiffeners

Graphite/Epoxy - Narmco 5208/ Thornel T-300

All other components Boron/Epoxy

This program was concerned with the design, manufacture, ground test, and flight test of an F-15 wing torque box utilizing boron/epoxy and graphite/epoxy composites. Initial design were progressively evaluated and developed through subcomponent and full scale static and fatigue tests.

F-4 Boron/Graphite/Polyimide Rudder - Boron/Polyimide and Graphite Polyimide)

The program was to design, fabricate and test a polyimide matrix composite rudder. The process was developed and the rudder successfully fabricated. It was static tested to failure at 400 percent of design limit load.

A-4 Horizontal Stabilator - Narmco 5206 Resin/Graphite Type II Fiber

Program demonstrates use of graphite/epoxy in aircraft primary structures. One stabilizer failed at 50 percent limit load in major joint to fuselage area. Redesign of second stabilizer is in progress. Estimated weight saving is 28 percent.

A-4 Wing Landing Flap - Boron/Epoxy - Graphite/Epoxy

The principal objectives of this program were to demonstrate the weight savings possible with the use of advanced composites and to compare the advantages of boron/epoxy with graphite/epoxy in the same application. The composite flaps are 6 ft. in span and 2 ft. in chord and have a constant cross section about 2 inches deep at the leading edge.

NORTHROP CORP.

F-5 Horizontal Stabilizer - Graphite/Epoxy

The composite horizontal stabilizer, designed to be a flightworthy primary structure and fully interchangeable with an existing metal stabilizer, has successfully completed a 16,000 hrs. (4 lifetimes) fatigue test. The stabilizer exhibited similar deflection characteristics to the production metal stabilizer and displayed more than adequate fatigue life. No degradation or fatigue damage was detected either during or after the test.

F-5E Trailing Edge Flap Graphite/Epoxy, Boron/Epoxy
A-9A Rudder PRD-49, E-Glass and Hybrid

The F-5E trailing edge flap and the A-9A rudder were selected as demonstration components. Preliminary conceptual studies for the components have been completed. Preliminary design of these components is currently in progress. Fabrication processes for these components, conventional autoclave, platen press, and vacuum bag, utilizing graphite/epoxy, boron/epoxy, PRD-49, E-glass and hybrid materials are being compared for cost reduction potential.

ROCKWELL INTERNATIONAL

F-100D Wing Outer Panel - Upper and Lower Skins - Boron/Epoxy

The structural elements chosen were the upper and lower skins of the F-100D wing panel. The skin is approximately 20 ft. in span, with chord dimensions of about 5 feet at the inboard end and 2 ft. at the wingtip. The panel thickness varies from about 1.6 inches at the inboard end to 0.1 inch at the tip. Estimates based on preliminary design drawings indicate a 21.9 percent weight saving over the comparable aluminum skins.

It was expected that current cure cycles would not be satisfactory because of the exotherm reaction resulting from the very thick sections. A testing plan was established to develop the proper cure cycle for the lower exotherm reaction.

B-1 Boron-Reinforced Longerons - Boron/Epoxy and Plastilock 731

This program is to design fabricate and test dorsal and lower longerons selectively reinforced with boron/epoxy for the aft fuselage of the B-1. This reinforcement is necessary to provide aeroelastic stiffness for support of the horizontal stabilizer. They are scheduled to be installed on the first B-1 air vehicle. The upper dorsal longeron is 47 1/3 feet in length. The main fuselage lower longerons are 27 2/3 feet in length. The lower inboard longerons are 19 1/2 feet long.
The boron/epoxy is bonded to the metal straps during the cocuring operation, and B. F. Goodrich Plastilock 731 was selected as the bonding adhesive. This selection was based on good lap shear strength, high peel resistance, and good cure flow characteristics.
441 pounds of boron/epoxy results in a weight reduction of 1185 pounds from the all metal design.

HELICOPTER AND VTOL SYSTEMS APPLICATIONS

BELL

UH-1 Tail Rotor Drive Shafts - Boron/Epoxy, Graphite/Epoxy

Systems analysis indicated a weight saving of approximately 28 percent. The driveshaft has a length between supports of 97.25 inches, and would be required to sustain an ultimate torque of 20,000 inch-pounds. Two full-length boron/epoxy and two full-length graphite/epoxy driveshafts were fabricated by Whittaker and shipped to Bell for structural and dynamic ground tests.

BOEING CO. VERTOL DIV.

HLH Main Rotor Synchronization Shaft - Combinations of glass, graphite and PRD-49 filaments

Because of a problem encountered in obtaining adequate low velocity impact resistance, the potential weight saving had been seriously eroded by the increase in tube weight from 141 pounds, for the all-graphite/epoxy system, to 233 pounds for the glass-graphite/epoxy system. The resulting approximately 12 percent weight reduction over the all metal design was not considered to be cost effective in view-of the potential field problems involved.

CH-47 Aft Main Rotor AGB Blades - Boron/Epoxy

The program was to develop the technology for the design and construction of the boron/epoxy blades. The program was completed and no flight restrictions were found. The blade performance and stability was excellent. Both the quantitative and qualitative impressions developed are extremely encouraging for the boron blade. A minimum increase of 3500 pounds in rotor limit gross weight capability in forward flight was projected for the CH-47C helicopter with Boron blades.

MODEL 222 TILT ROTOR SYSTEM - Hybrid of Fiberglass/Epoxy and Boron/Epoxy

A key factor in the development of the Model 222 tilt rotor hytrid advanced composite propeller system was the use of materials with a high ratio of fatigue strength to elastic modulus. The ability to combine and tailor the fiberglass and boron composites to attain desired structural properties was a significant factor in their selection. The design, fabrication, and fatigue tests are completed. The feasibility of using hybrid composite fibers in a single system was demonstrated.

SIKORSKY

CH-54B Tail Cone - Hybrid of boron/epoxy and aluminum

Phase I study confirmed the fact that boron/epoxy - reinforced stringers can be fabricated and that their static and fatigue strength is adequate to meet the requirements of the CH-54B.
The boron/epoxy reinforced tail cone was fabricated and Fokker bond ultrasonic tested. The installed tail cone was ground checked and found to be 10 percent higher on vertical bending frequency than the previous production aluminum reinforced airframe.
Flight testing confirmed that the tail cone with the boron/epoxy-reinforced tail cone eliminated any significant dynamic coupling between the rotor and airframe.

CH-54B TAIL SKID GEAR TUBES - Boron/Epoxy

Boron/epoxy tubes were designed to replace the conventional swaged aluminum members of the CH-54B tail skid gear. Two series of tests were conducted and showed the tubes to have a 50 percent margin of safety over the design load requirements. The tubes were installed on the helicopter and released for service.

S-61 Tail Rotor Blades - Hybrid of Boron-glass/epoxy

The purpose of the program was to flight test a vital dynamic component made of boron composite. The tail rotor blades of the S-61 helicopter was selected.
A full set of five rotor blades was whirl-tested at rated speed and found to be flightworthy. A full set of boron-glass/epoxy blades was then flight tested.

S-61 Drive Shaft - Boron/Epoxy

The higher structural efficiency of boron/epoxy compared with present metal shafts promises weight savings of more than 50 percent in shafting. The tail rotor drive shaft system for the S-61 aircraft consists of a sandwich structure in which two boron laminates were separated by a low-density core of foam or honeycomb. An integral end fitting in which the fibers continued into the end was developed. Testing of the integral end fitting showed it to be structurally sound.

MISSILE AND SPACE SYSTEMS APPLICATIONS

AEROJET GENERAL CORP.

Filament-Wound Pressure Vessels - Cryogenic Service - Boron/Epoxy

Type 304 corrosion resistant steel (.006 inch foil) with boron/epoxy filament overwind. The foil liner is cryogenically stressed for strain

compatibility. The vessels displayed very low biaxial strains and high strength to weight ratios. Such low strains permit high specific strength metal liners to work elastically up to the ultimate stress of the boron filament windings.

BOEING CO.

Missile Interstage Structure with Hat Stiffeners and Rings - Graphite/Epoxy

The objective of this program was to investigate fabrication processes and their relationship to design, fabrication and test of high-modulus graphite-reinforced composite structural elements representative of a Minuteman III missile interstage structure.
The cylindrical elements were tested under combined axial compression and external pressure and developed 200 percent of design load. The various terminal joint configurations demonstrated that adhesive bonding can be used to transmit high loads into graphite/epoxy composite laminates. However, the cohesive strength of the composite material was the joint-strength limiting factor rather than the strength of the adhesive itself.

Reinforced Pressure Vessels - PRD-49/Epoxy with Boron/Epoxy

Construction has been evaluated with pressurized filament wound cylinders and small pressure vessels with good success. Advancement of the technology to applications of completely wound pressure vessels of flightweight construction is pending. One difficulty encountered with the material, i.e., its relatively low specific compression strength, has promise of being overcome by combining the PRD-49 fibers in a sandwich - composite construction having the outer layers of the laminate reinforced with fibers of higher compressive strength.

Shuttle Titanium Clad Boron/Epoxy Shear Web

Fabricate and test titanium clad boron/epoxy laminate as a shear web with vertical stiffener members locally reinforced with boron/epoxy laminate. Also sandwich web configuration with titanium clad boron/epoxy facings. Preliminary material properties studies completed. Sections of flight-weight shear web construction fabricated and statically tested.

Shuttle Orbiter Reinforced Fuselage Frame - Boron/Epoxy

Program is to evaluate the feasibility of designing a Space Shuttle Orbiter titanium fuselage frame reinforced with boron/epoxy composite laminates. Boron/Epoxy-reinforced frame is approximately 25 percent lighter than an optimized all-titanium frame. The material that was selected for the design of the metal frame is 6 Al - 4V titanium. The composite-reinforced counterpart will incorporate the use of Rigidite 5505/4 laminate. Metalbond 329 is the adhesive for assembling the various reinforced frame components.

GENERAL ELECTRIC CO.

Reentry Vehicle Composite Frustrum - Boron/Epoxy (Filament-wound)

The objectives were to develop a design, perfect fabrication techniques, and test an advanced composite structure representative of a reentry vehicle configuration which incorporates an insulation system to maintain the load-carrying structure within acceptable strength to temperature limits. The configuration chosen incorporated an ablative-type nose cap, a boron/epoxy filament - wound conical frustum structure with an overlaid state-of-the-art thermal shield and an aluminum hemisphere structure with a bonded thermal shield.

GOODYEAR AEROSPACE

Millimeter Wave Length Antennas - Material - Graphite/Epoxy

Paraboloid shaped antenna of sandwich type construction having aluminum core and graphite/epoxy faces. Graphite/Epoxy feed support struts were fabricated. The 30 inch diameter antenna was completed, thermal cyclic tested and exposed to ultraviolet and gamma radiation. All objectives of the program were met.

Grumman

Tubular Strut - Shuttle Thrust Structure - Boron/Epoxy

A pin-jointed tubular truss, thrust structure replaced a titanium tubular truss. Program completed with one third scale truss members and segment of the assembled truss. Structure failed at 121 percent of ultimate design load with a 32 percent weight saving.

Jet Propulsion Lab.

Filament-Wound Rocket Motor Chambers - ATS Satellite - Boron/Epoxy

Program to design, fabricate and test advanced composite rocket motor chamber for potential replacement of the 6Al-4V applications on the ATS Technology Satellite. Chambers were successfully fabricated and passed the hydrostatic proof tests.

MCDONNELL DOUGLAS ASTRONAUTICS CO. WEST

Shuttle Landing Gear Door Assembly - Graphite/Epoxy - Aluminum Honeycomb Core Sandwich

Originally designed rib-stiffened titanium panel was selectively reinforced with graphite/epoxy laminates. The subsequent redesign is to use a sandwich with aluminum honeycomb core and graphite/epoxy faces. The design, specimen test and fabrication of landing gear door segments are complete. Static and environmental testing are pending.

NORTHROP CORP.

Tubular Strut - Graphite/Epoxy and Glass/Epoxy

Design consisted of a platform support strut element with combinations of glass cloth and Thornel 50 graphite. The program is completed. Light weight fittings were designed. There was a 46 percent weight saving on the tube and 25 percent on the end fittings.

ROCKWELL INTERNATIONAL/SPACE DIV.

Cryogenic Support Struts - Boron/Epoxy

Design consisted of a network of pin-ended struts connecting an insulating shroud to the forward skirt of a tank. The following conclusions resulted from this program: (1) boron/epoxy is an excellent material for structure requiring a low thermal conductivity and high stiffness or stability under compressive load; (2) no serious mechanical or physical property defects were noted in the cryogenic temperature regime; (3) the low temperature tolerance of adhesively bonded structures to eccentricities requires further study of adhesive bonds for cryogenic applications; and (4) titanium metal skins in lieu of steel are desirable for low-temperature reinforcement of boron/epoxy because of a good coefficient of expansion match.

TRW/SYSTEM DIV.

Model 35 Spacecraft Strut - Boron/Epoxy

Initiated by TRW Systems to further the state-of-the-art of lightweight structures. Effort was to design, fabricate, and evaluate boron/epoxy composite struts as a substitute for aluminum counterparts used in the Model 35 Spacecraft. Although the end fittings and attachments were not optimized, a weight savings of approximately 30 percent was realized over the aluminum counterpart.

Pioneer 10 Spacecraft Tubular Struts - Boron/Epoxy

The tubular support struts were designed to carry pure axial compression and tension loads and were designed with titanium clevis type end fittings for pin attachment to the platform and adapter cylinder. The stepped end fittings were bonded to the tubular section. Prototype struts successfully subjected to a series of tests including compression and tension cycling to verify the analytical predictions. Production struts have passed all qualification tests and were flown on the Pioneer 10 Spacecraft launched in March 1972.

Extra-High Frequency (EHF) Antennas - Graphite/Epoxy

The first 4-foot reflector was made in 1969, and the first 9-foot reflector of upgraded materials was completed in 1971 and underwent environmental tests during 1972. The test data from these projects provided the stimulus for initiating a further improved 108-inch antenna. This incorporated a number of improvements, including: (1) higher temperature (350°F) material system, (2) lower thermal expansion, and (3) more isotropic biaxial properties (not including thickness direction) of laminates over the entire contour. The antenna has been subjected to extensive thermal distortion, extreme temperature cycling, and acoustic testing. It met or exceeded all of the test objectives.

Conclusions

1. Materials shown in the tables, when molded using low pressure laminating methods, are suitable for use in aerospace primary structural components where high stiffness and strength-to-weight ratios are required.

2. The data tabulated herein reflect the actual properties of the materials covered, since they are the averages of the data points recorded from the test programs described in references 1 thru 23.

3. At this stage in the development of advanced composites technology many of the available data still do not represent the results of a large population of tests.

4. There is sufficient data available to categorize the broad characteristics of a brand name material. (Tables III, IV, V, VI, and IX)

5. The epoxy matrix composites can be used in applications up to 350°F. The greatest amount of data are available for these materials. (Tables III, IV, and IX)

6. Material specifications must be established for each constituent of composite prepregs and for the prepreg itself. This is necessary since there is a great variance in the final properties of materials under the same brand name. (References 2, 9, 14, and 23)

7. Graphite fiber composites are excellent for dimensionally stable structures because of their low thermal expansion coefficients. (Table IX)

8. Components or structure fabricated from Thornel 300/Narmco 5208 graphite/epoxy prepregs have premium strength properties. (Tables III and IX)

9. The graphite/polyimide (710) composites and processing are compatible with all common graphite fibers except the ultra high strength fibers. (Tables V and VI)

10. The polyimide resin matrix composites are useful over the temperature range of -67 to 550°F continuous service. (Tables IV and V)

11. Laminates up to two inches thick can be fabricated from the graphite/polyimide system. (Reference 14)

12. It is concluded from the current advanced composite hardware programs section of this report that there are three distinct methods of advanced composite utilization. Each method has different payoffs, but all have definite advantages over standard all-metal systems. The three methods are:
    (a) Conceptual composite design
    (b) Element-by-element substitution of composite for metal.
    (c) Selected reinforcement of metal elements with composites.

13. Twenty-nine current advanced composite hardware programs completed in the United States and described in this report are 100 percent successful in that they demonstrated a significant weight saving or improved structure from an increased fatigue life or design limit load standpoint. The balance of thirteen programs described herein have not been completed.

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